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A temperature survey of the region aft of a cooling orifice of an air film cooled metal surface exposed to high temperature and high velocity gases

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A TEMPERATURE SURVEY OF THE REGION AFT
OF A COOLING ORIFICE OF AN AIR FILM COOLED METAL
SURFACE EXPOSED TO HIGH TEMPERATURE AND HIGH VELOCITY GASES

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TABLE OF CONTENTS

Acknowledgements	ii
Table of Contents	iii
Summary	1
Introduction	3
Test Equipment	6
Test Procedure	12
Results and Discussion	14
Conclusions	18
Bibliography and References	21
Nomenclature	22
Tables I to IV	23
Figures 1 to 11	27
Sample Calculations	36

SUMMARY

The object of this investigation was to determine the temperature pattern in the region downstream from a cooling air orifice of an air film cooled simulated turbine blade exposed to high velocity and high temperature gases. It was particularly desired to determine the pattern within the boundary layer of the flow over the blade since this pattern would indicate the degree of mixing and the extent to which the cool film remained on the blade surface.

Tests were made on a static rig with a flat plate used in place of a regular turbine blade, and with cooling air introduced normal to the blade surface. The Mach of the hot gas flow was approximately .5 at a total temperature of 1215° F. The range of temperatures in the pattern was approximately 1090° to 1215° F. The temperature probe consisted of a small bead thermocouple attached to a micrometer barrel. The probe was mounted on a frame which rested on top of the test section above the blade so that the probe extended through a slot into the test region, being positioned vertically by the micrometer barrel, while clamped at the respective horizontal locations.

The following observations were made:

Temperature patterns were obtained within the boundary layer, showing a temperature gradient through the boundary layer that was proportional to the weight flow of cooling air. The lowest temperatures were found to be about 3/8" downstream from the cooling air orifice. Vertically, the effectiveness of the cooling was negligible above .15" above the blade, and horizontally, the effectiveness appeared to be decreasing rather rapidly at a distance of 1" downstream from the orifice.

INTRODUCTION

The need for increasing turbine inlet temperature in aircraft gas turbines in order to increase efficiencies and outputs has been recognized for several years. Turbine theory shows that for a given capacity or airflow the power per pound of air is proportional to turbine inlet temperature. Since the temperature increase is limited by the high temperature strength of the blade materials, considerable effort has been expended in trying to find a suitable means of cooling the blades so that they may operate at a temperature below that of the surrounding gases. Ref. 2 discusses the need for the higher inlet temperatures and points out that the successful method must take into account fabrication problems and stress considerations of the completed blade as well as degree of cooling.

Up to now analytical investigations have not been very successful. The mechanism of flow in the boundary layer surrounding a turbine blade is not yet sufficiently understood to make accurate quantitative comparisons. The convection heat transfer coefficient from blade to coolant is needed for a theoretical analysis. This involves the flow

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Up to now analytical investigations have not been very successful. The mechanics of flow in the boundary layer surrounding a turbine blade is not yet sufficiently understood to make accurate quantitative predictions. The convection heat transfer coefficient from blade to coolant is needed for a theoretical analysis. This involves the flow of heat through a boundary layer. The equations for temperature variation in a laminar boundary layer have been set up but involve nine equations, five of which are differential equations, with nine unknowns. And for turbulent boundary layers, the equations are more difficult. Since solutions are impractical the field of experimental investigations must be turned to.

One of the more practical methods investigated is film cooling, an attempt to substitute a boundary layer of cool air over the blade surface in order to inhibit the heat transmission from the hot gases to the blade. While this method appears to be worthy of further development, it has been observed in tests to date (Ref. 6 and Ref. 7) that the effectiveness of the cooling air film decreases rapidly downstream from the cooling air orifice. This is not good for turbine blade cooling as the blades must be very thin in the area near the trailing edge so that cooling air would have to be introduced well forward of the trailing edge.

Since conditions within the boundary layer are of prime importance in this method of cooling the object of this investigation was to determine the temperature patterns in and near the boundary layer of a film cooled blade with

the hope that a study of these patterns might give more insight into the phenomena of film cooling.

TEST EQUIPMENT

Test Blade

It was decided to use the same blade that was used in the investigation described in Ref. 7 since it met the requirements for this investigation. Fig. 7 is a picture of the blade. Use of this blade was considered particularly advantageous since it gave a flat surface and it was desired to isolate the mixing tendencies of the hot gas and cooling air and not include the effects of blade twist or of turning of the flow on the mixing.

The blade was made of thin, mild steel, was flat sided and had zero camber. Only the first of the five rows of cooling air orifices was used in the test. There were 33 holes to a row, each hole .04" in diameter. Only the top surface of the blade was cooled, the cooling orifices being normal to the blade surface.

The cooled surface of the blade was made equal in area to the total area of an actual J33 turbine blade. This feature allowed a rough comparison to be made to the cooling of an actual turbine.

Temperature probe

The temperature probe is shown in Fig. 8 and is shown in position on the test section in Fig. 10. The requirements for the probe were that its horizontal, vertical, and lateral position with reference to the test blade be known and that it offer a minimum disturbance to the flow over the blade. A considerable proportion of the time spent on this thesis was devoted to the construction of the probe.

The ends of a "U" shaped frame were bolted to a flat plate. The stationary element of a micrometer barrel was held in place in the top of the frame by a set screw and the moving element extended down and acted against a thermocouple holding piece which moved up and down on guide rods between the top of the frame and the base plate, being held tight against the micrometer barrel by spring pressure. The thermocouple wires just above the bead were in a ceramic insulator which was held in place in the thermocouple holding piece by another set screw with the bead end extending through a hold in the base plate. In effect then the bead was constrained to follow the movement of the micrometer barrel and the micrometer reading could be taken for any point.

Another flat plate somewhat larger than the probe base plate was bolted to the top of the test section and had metal guides on each side so that the probe could be moved fore and aft on it but not laterally. A scale was scribed on one guide and a scribe mark placed on the probe base plate so that its horizontal location could be read. A wing nut set screw was threaded through one guide to bear against the probe base plate and hold it at any specified horizontal station. A slot in the center of the plate and the test section wall, paralled to the direction of the gas flow, accomodated the extended probe.

In order to positively establish the point of contact of the probe with the blade a 3 volt potential was put across the probe and the blade with a small light and a switch in the circuit. With the switch on, the light went on when the probe contacted the blade and when the switch was off there was no interference with the temperature reading.

Temperature Recording System

All thermocouples used were iron-constantan and were read on a Brown Recording Potentiometer having a scale from 0 to 1600° F., readable to ± 5° F. Thermocouple beads

were formed by the arc process using a mercury pool covered with lubricating oil. Burner air inlet, cooling air inlet to rotameter, survey probe, and test section reference temperature were recorded. The latter is referred to as reference temperature even though it was an attempt to get test section total temperature. A radiation shielded total temperature probe was used but it consistently read about 80° F. lower than bare thermocouples in the vicinity. After carefully checking the set up and noting that other students doing thesis work with different but comparable probes had the same difficulty it was concluded that since the probe extended no more than 1" into the hot gas duct, the conduction from the shield itself to the outside was the probable cause for the error.

In conjunction with this test section temperature probe, test section static and total pressure probes were used so that the Mach number of the flow over the blade could be determined.

Hot Gas System

Fig. 11 is a picture of the test cell and shows the basic elements of the hot gas system. Fig. 9 is a picture

of the control panel for the test cell. In the test cell was a Lycoming Model O-435-T air cooled engine rated at 162 h.p. at 2800 R.P.M. driving the air compressor which was a 7.48 to 1 gear ratio supercharger from an Allison V-1710 aircraft engine. The air delivered by the supercharger to a large manifold was ducted to a single Allison J33-A-17 turbojet engine combustion chamber. Combustion was started by a spark-ignited acetylene flame and combustion temperatures were controlled by the burner fuel pump bypass, for regulating fuel flow, and by the engine R.P.M. which controlled supercharger flow rate.

The hot combustion gases were then ducted through the test section to an exhaust manifold.

Air flow to the burner was measured at an orifice on the intake side of the compressor. The orifice was 5.6" in an 8" pipe and radius type static pressure leads were connected to a water manometer at the control panel.

Fuel flow to the burner was measured by a Fischer and Porter "Flowrator" tube # 5A-60 on the control panel.

Cooling Air System

Cooling air was supplied from the compressed air system of the Mechanical Engineering Building. Pumping capacity of the system was greater than the maximum flow rate. The cooling air flow rate was determined from a Fischer and Porter "Flowrator" # 5A-25 tube calibrated at 100° F. and 14.7 psia. Cooling air temperature and pressure were measured at the entry to the flowrator for converting the measured flow rate to the actual conditions of temperature and pressure.

TEST PROCEDURE

The requirement of the test procedure was that conditions in the test section remain constant during a given run. This requirement was met by maintaining the test section reference temperature and the burner airflow constant. The procedure was to maintain the differential pressure across the burner air orifice constant by varying the R.P.M. of the engine with the throttle. The reference temperature was held constant by varying the burner fuel flow as measured by the fuel rotameter. In both cases the adjustments during a given run were very slight and the test conditions were apparently accurately maintained constant.

The desired cooling air flow was maintained constant by holding a given air rotameter reading for a given run.

The procedure for getting the probe temperatures was for the author to remain in the test cell and after test conditions were established to establish contact between the probe and the blade and then to make predetermined settings above the blade using hand signals to indicate to the

control personnel when to read the Brown Recorder. The same procedure was repeated for each selected horizontal station along the blade. Tables I, II, and III list the observed data for all runs.

RESULTS AND DISCUSSION

The results of the investigation are presented in Tables I to IV and the curves of Figs. 1 to 6, inclusive. Figs. 1 to 3 show the temperature profiles at each horizontal station for the three runs while Figs. 4 to 6 show the temperature patterns above the blade for the three runs.

There were some limitations to the accuracy of results but none that was not expected. Ignoring small radiation and conduction losses, the survey temperatures still were not exact because the recovery factor for a thermocouple in a high velocity air stream is approximately 85 per cent (Ref. 4). For the test conditions of combustion gases at a Mach of .5, T_S/T_O is .96 (Ref. 8). So an error of .6 of one per cent would be expected which would be 9° error at 1660° R. However, this relatively constant error would have no effect on the temperature profiles or patterns.

Because of the impossibility of measuring temperatures at the mathmatical concept of a point, no readings were actually taken at the surface of the blade. Information from Ref. 10 would indicate that the effective junction of the thermocouple bead would be approximately the

diameter of the wire, .03", above the tip of the bead so that an error of about .03" was probably present in all specified vertical positions. This did not affect the validity of the results, however, since the turbulent boundary layer of the flow over the blade was about .07" (see Sample Calculations), so that temperature readings were definitely obtained in the boundary layer.

Figs. 1 to 3 show the gradient caused by the cooling air. The fact that the gradient was caused only by the cooling air, and not by conduction from the blade or some other reason, was established by making surveys with no cooling air at all horizontal stations on run #2 and at two horizontal stations on run #3. In each case the temperature throughout the boundary layer was found to be constant and to be the same as the free stream temperature.

There was virtually no cooling effect more than .15" above the blade. Figs. 1 to 3 show the profiles to be nearly vertical lines above that height. This would seem to be a good point for the film cooling method since it indicates that much of the cooling air is held near the blade even when injected normal to it. This result is not in agreement with the suggestion in Ref. 3 that because of the low diameter of the wires, .020", above the lip of the hood so that an error of about .002 was probably present in all vertical positions. It is felt the effect [illegible] the majority of the readings, however, since the [illegible] [illegible] of the film over the plate was about .007" [illegible] [illegible] [illegible]), so that temperature readings were [illegible] [illegible] in the boundary layer.

Figs. 4 to 8 show the profiles caused by the heating wire. The fact that the profile was caused only by the heating wire and not by conduction from the sides or some other reason, was established by making traverses with no heating wire at all [illegible] [illegible] [illegible] of the horizontal [illegible] [illegible]. In each case the temperature throughout the boundary layer was found to be [illegible] to be the same as the free stream temperature.

There was difficulty in [illegible] [illegible] when [illegible] .050 [illegible] the plate. Figs. 4 to 8 show the profiles to be nearly vertical [illegible] [illegible] [illegible]. This would seem to be a good [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible] [illegible]. This result is not in agreement with [illegible] [illegible] [illegible] 2 [illegible] [illegible] of the [illegible] momentum of the boundary layer cooling air passes through it almost undisturbed. However, the reasoning may well apply to the laminar sublayer of the boundary layer where no measurements were possible.

A comparison of Figs. 5 and 6 shows the effect of increasing the cooling air flow for a given burner air flow. Increasing the cooling air flow from .36 lb/min to .56 lb/min increases the maximum cooling effect from 60° to 120° F. and changes the shape of the temperature pattern in the region near the orifice. The smaller weight flow appears to be swept along by the hot gases while the larger flow has considerably more penetration.

Figs. 4 to 6 all show the point of maximum cooling to be about 3/8" downstream from the cooling air orifice. There seems to be no positive explanation for this result but it is possible that at this point an optimum relation exists between the factors of mixing of that part of the cooling air that penetrated the boundary layer and convection heating of that part that did not penetrate the boundary layer but was carried downstream with it. As you go further downstream the cooling effect falls off. This rapid decrease in cooling effectiveness downstream of the orifice has been

noted in Refs. 6 and 7. However, the point of maximum cooling has been seen to be 3/8" downstream, a reasonable distance when thought of in terms of the dimensions of a turbine blade. Further, the increased penetration of the greater cooling air flow shows that it is quite possible that the point of maximum cooling might be moved further downstream with some particular combination of angle of entry, velocity, and rate of cooling air flow. If the point of maximum cooling could be moved further downstream the method of film cooling would appear to be very promising and for that reason it is believed that additional tests should be made with variable cooling air velocities, entry angles, and weight flows.

The area of the cooled surface of the test blade was equal to the total area of a J33 turbine blade so that a rough comparison to an actual blade could be made. Results show that the maximum cooling air flow used in this investigation compares to 1.6% of compressor air for comparable cooling of the J33 engine (see Sample Calculations). Ref. 2 indicates that up to 7% of compressor air is reasonable for cooling but it must be remembered that the above comparison is rough since cooling is a function of other variables, not considered here, such as Reynold's number.

CONCLUSIONS

Because of the limitations imposed by the use of a static rig and a simulated turbine blade no conclusions can be made regarding the temperature pattern in a film cooled turbine but the following qualitative conclusions were reached for the equipment used and the conditions of the test:

1. Where no cooling air is introduced the temperature within the boundary layer is constant and is the same as the temperature of the free stream.

2. When cooling air is introduced there is a temperature gradient through the boundary layer. This is true for a distance of at least 1" aft of the cooling air orifice. Since the gradient begins to fall off at about 3/4" downstream from the orifice it would appear that the effect of the cooling air is very small beyond 1½" to 2" downstream.

3. The lowest temperatures were found to be about 3/8" downstream from the orifice, very near the blade surface. An explanation for this might be that at that dis-

tance downstream an optimum relation exists for the factors of mixing of that part of the cooling air that penetrated the boundary layer and convection heating of that part of the cooling air that did not penetrate the boundary layer but was carried downstream with it.

4. The temperatures within the boundary layer varied with the rate of cooling air flow increasing rates of cooling air giving lower temperatures. However, for a given installation the larger cooling air rates give greater penetration of the stream so that for the practical use of cooling a turbine blade there would be a maximum efficiency beyond which the increased velocity and density of the cooling air would give deep stream penetration but possibly poor film cooling.

5. The cooling effect was negligible above .15" above the blade.

6. Contrary to the reasoning that because of the low momentum of the boundary layer the cooling air passes through it almost undisturbed, the results show that part of the cooling air is swept along in the low velocity boundary layer. A boundary layer thickness of .07" is

typical while the probe actually recorded temperatures to as little as .03" above the blade and indicated a definite gradient there. However, it is quite possible that the laminar sublayer of the boundary layer did offer little resistance to the passage of cooling air through it.

7. The temperature patterns cover a range of about 1090° F. to 1215° F. for a burner air to cooling air ratio of 132 lb/min to .56 lb/min or 236 to 1. A rough comparison to a J33 jet engine would indicate that a 1.6% of compressor air bleed off for cooling would be necessary to give the same degree of cooling to that engine, ignoring dynamic effects.

8. While the temperature reduction was not exceptional in this method of blade cooling, the temperature patterns do show: 1) a cooled boundary layer for a reasonable distance downstream; 2) the lowest temperature to be somewhat downstream from the orifice; and 3) a pronounced effect on the patterns with varying rates of cooling air flow. It is therefore concluded that more tests of this nature should be made with various entry angles for the cooling air and with various sized slots and orifices in the blades.

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NOMENCLATURE

Bar.	Barometer	Inches Hg. Abs.
P_S	Test Section Static Pressure	Inches Hg. Abs.
P_T	Test Section Total Pressure	Inches Hg. Abs.
M_{TS}	Test Section Mach Number	--
M_B	Mach at Test Blade	--
ΔP_{BA}	Burner Air Pressure Differential	Inches Water
T_{BA}	Burner Air Temperature	^{o}F.
T_{CA}	Cooling Air Temperature	^{o}F.
W_{BA}	Weight Flow of Burner Air	LB/SEC
W_{CA}	Weight Flow of Cooling Air	LB/MIN
W_{BF}	Weight Flow of Burner Fuel	LB/HR
P_{CA}	Cooling Air Pressure	LB/IN^2
Q_{ROT}	Cooling Air Rotameter Reading	C.F.M.
Re.	Reynolds Number	--
δ	Boundary Layer Thickness	Inches
γ	Specific Heat Ratio	--
μ	Viscosity	LB/FT.SEC.

TABLE I

DATA SHEET

TEST RUN # 1 – LOW COOLING AIR FLOW (.363 LB/MIN)

DATE: 7/25/51
TEMP: 85°F
BAR: 29.23" Hg.

STATION	FUEL PRESS. P_F (psig)	ENGINE RPM	TEST SECTION STATIC PRESS. P_S ("Hg ABS)	BURNER AIR PRESS DIFF. ΔP_{BA} ("H_2O)	BURNER FUEL FLOW W_B (LB./HR.)	TEST SECTION TOTAL PRESS. ΔP_T ("Hg)	REFERENCE TEMP T_R (°F)	BURNER AIR TEMP. T_{BA} (°F)	COOLING AIR TEMP T_{CA} (°F)	COOLING AIR PRESS P_{CA} (psig)	COOLING AIR ROTAM. Q_{ROT} (CFM)
1	103	2520	31.5	15	125	6.2	1130	90	90	12	3.8
2	103	2530	31.5	15	125	6.3	1130	95	95	12	3.8
3	100	2530	31.5	15	122	6.3	1130	95	95	12	3.8
4	100	2530	31.5	15	123	6.4	1130	95	95	12	3.8
5	100	2530	31.45	15	120	6.2	1130	95	95	12	3.8
6	100	2530	31.45	15	122	6.2	1130	95	95	12	3.8

SURVEY PROBE TEMPERATURE

VERTICAL HEIGHT ABOVE BLADE	HORIZONTAL STATION 1 AT ORIFICE	2 1/8" AFT	3 1/4" AFT	4 3/8" AFT	5 3/4" AFT	6 1" AFT
FLUSH	1160	1170	1140	1130	1135	1120
.005"	1180	1180	1150	1140	1145	1140
.010	1200	1195	1160	1140	1145	1150
.015	1220	1200	1165	1150	1150	1160
.020	1240	1200	1170	1160	1155	1165
.025	1250	1205	1175	1170	1155	1165
.050	1260	1240	1190	1175	1175	1170
.100	1260	1250	1200	1190	1220	1185
.150	1255	1260	1200	1190	1220	1190
.200	1265	1255	1200	1190	1220	1190
.30	1265	1255	1200	1190	1215	1190
.40	1260	1258	1200	1190	1205	1185
.50	1260	1255	1200	1190	1205	1185

TABLE II

DATA SHEET
TEST RUN. #2 - LOW COOLING AIR FLOW

DATE: 7/27/51
BAR: 29.23
TEMP: 84°F

HORIZONTAL STATION	FUEL PRESS. P_F (psig)	ENGINE RPM	TEST SECTION STATIC PRESS. P_S ("HG.)	BURNER AIR PRESS. DIFF. ΔP_{BA} ("H_2O)	BURNER FUEL FLOW W_B (LB/HR)	TEST SECTION TOTAL PRESS. ΔP_T ("HG)	REF. TEMP T_R (°F)	BURNER AIR TEMP. T_{BA} (°F)	COOLING AIR TEMP. T_{CA} (°F)	COOLING AIR PRESS. P_{CA} (psig)	COOLING AIR ROT. $Q_{ROT.}$ (CFM)
A. 1 AT ORIFICE	98	2570	31.6	15	122	6.9	1130	90	85	0	0
2 1/8" AFT	98	2580	31.6	15	122	6.9	1130	90	85	0	0
3 1/4" AFT	98	2580	31.6	15	121	6.9	1130	90	85	0	0
4 3/8" AFT	95	2530	31.4	15	117	6.5	1130	95	90	0	0
5 5/8" AFT	95	2530	31.5	15	116	6.5	1130	95	90	0	0
6 1" AFT	95	2530	31.5	15	115	6.5	1130	95	85	0	0
B. 1	98	2580	31.6	15	122	6.9	1130	90	85	12	3.75
2	98	2580	31.6	15	122	6.9	1130	90	85	12	3.75
3	98	2580	31.6	15	122	6.9	1130	90	85	12	3.75
4	95	2530	31.5	15	116	6.5	1130	95	90	12	3.75
5	95	2535	31.5	15	116	6.5	1130	95	85	12	3.75
6	95	2530	31.5	15	115	6.5	1130	95	85	12	3.75

A. NO COOLING AIR
B. .363 LB/MIN COOLING AIR

SURVEY PROBE TEMPERATURE

VERTICAL HEIGHT ABOVE BLADE	HORIZONTAL STATION 1 AT ORIFICE	2 1/8" AFT	3 1/4" AFT	4 3/8" AFT	5 5/8" AFT	6 1" AFT
	A. NO COOLING AIR					
FLUSH	1210	1210	1220	1200	1200	1200
.005"	1215	1218	1220	1200	1202	1200
.010	1215	1215	1220	1200	1200	1200
.015	1210	1210	1218	1200	1195	1195
.020	1210	1210	1218	1200	1195	1195
.025	1215	1210	1218	1200	1195	1195
.050	1220	1215	1215	1200	1198	1195
.100	1220	1210	1218	1205	1198	1200
	B. WITH LOW COOLING AIR FLOW					
FLUSH	1175	1170	1165	1160	1145	1150
.005"	1200	1195	1180	1170	1150	1150
.010	1205	1195	1190	1175	1155	1155
.015	1205	1195	1190	1180	1158	1155
.020	1210	1198	1195	1185	1165	1160
.025	1210	1200	1198	1185	1170	1162
.050	1215	1205	1205	1190	1182	1180
.100	1215	1205	1215	1195	1190	1185
.150	1215	1205	1215	1195	1185	1190
.200	1218	1203	1218	1200	1180	1192
.300	1210	1205	1215	1200	1185	1190

TABLE III
DATA SHEET
TEST RUN # 3 — HIGH RATE COOLING AIR FLOW (.563 LB/MIN)

DATE: 7/30/51
TEMP: 85 °F
BAR: 29.13

HORIZONTAL STATION	FUEL PRESS. P_F (psig)	ENGINE RPM	TEST SECTION STATIC PRESS. P_S ("HG.)	BURNER AIR PRESS DIFF. ΔP_{BA} ("H_2O)	BURNER FUEL FLOW W_B (LB./HR)	TEST SECTION TOTAL PRESS. ΔP_T ("HG.)	REF. TEMP. T_R (°F)	BURNER AIR TEMP. T_{BA} (°F)	COOL AIR TEMP T_{CA} (°F)	COOL AIR PRESS. P_{CA} (psig)	COOL AIR ROT. $Q_{ROT.}$ (CFM)
A. 1 AT ORIFICE	98	2530	31.5	15	120	6.2	1130	90	90	0	0
5 1" AFT	97	2530	31.5	15	118	6.2	1130	90	80	0	0
B. 1 AT ORIFICE	98	2540	31.5	15	120	6.2	1130	90	80	22	4.95
2 1/8" AFT	98	2540	31.5	15	120	6.2	1130	90	80	22	4.95
3 3/8" AFT	98	2540	31.5	15	120	6.2	1130	90	80	22	4.95
4 5/8" AFT	96	2540	31.5	15	118	6.2	1130	90	80	22	4.95
5 1" AFT	97	2530	31.5	15	118	6.2	1130	90	80	22	4.95

A. NO COOLING AIR
B. .563 LB/MIN COOLING AIR

SURVEY PROBE TEMPERATURES

VERTICAL HEIGHT ABOVE BLADE	HORIZONTAL STATION 1 AT ORIFICE	2 1/8" AFT	3 3/8" AFT	4 5/8" AFT	5 1" AFT	6
		A. NO COOLING AIR				
FLUSH	1210				1200	
.005"	1210				1200	
.010	1210				1200	
.015	1205				1200	
.020	1205				1200	
.025	1205				1200	
.050	1210				1203	
.100	1205				1200	
		B. HIGH RATE COOLING AIR FLOW				
FLUSH	1170	1120	1090	1095	1108	
.005"	1190	1140	1100	1100	1110	
.010	1195	1140	1103	1105	1115	
.015	1195	1140	1110	1110	1120	
.020	1195	1160	1118	1115	1120	
.025	1198	1165	1120	1120	1128	
.050	1200	1165	1145	1140	1150	
.100	1203	1190	1170	1170	1180	
.150	1200	1190	1190	1190	1195	
.200	1200	1195	1190	1190	1195	
.300	1200	1195	1190	1190	1195	

TABLE IV

REDUCED TEST DATA

	Run 1	Run 2	Run 3
Bar	29.23	29.23	29.13
P_S	31.5	31.5	31.5
P_T	35.53	36.03	35.3
M_{TS}	.44	.46	.42
M_B	.51	.53	.48
ΔP_{BA}	15	15	15
T_{BA}	95	92	90
W_{BA}	2.24	2.24	2.24
T_{CA}	95	85	90
P_{CA}	12	0 and 12	0 and 12
W_{CA}	.363	0 and .363	0 and .563
W_{BF}	125	120	120
F/A Ratio	.0155	.0149	.0149
*% Burner Air for Cooling	1.04	1.04	1.60
Re.	1.38×10^5	1.38×10^5	1.38×10^5

*Based on J33 turbojet engine.

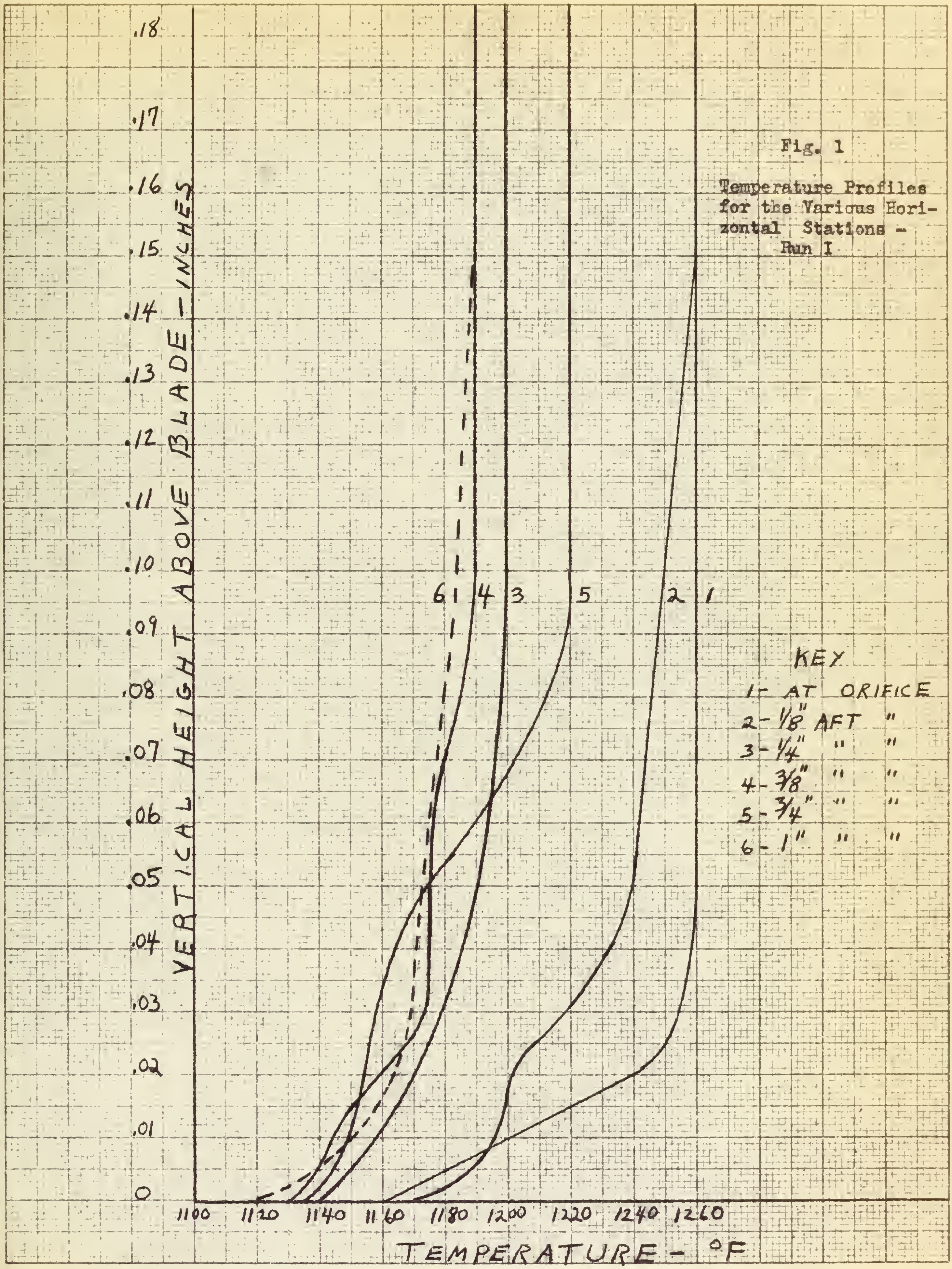


Fig. 1

Temperature Profiles for the Various Horizontal Stations - Run I

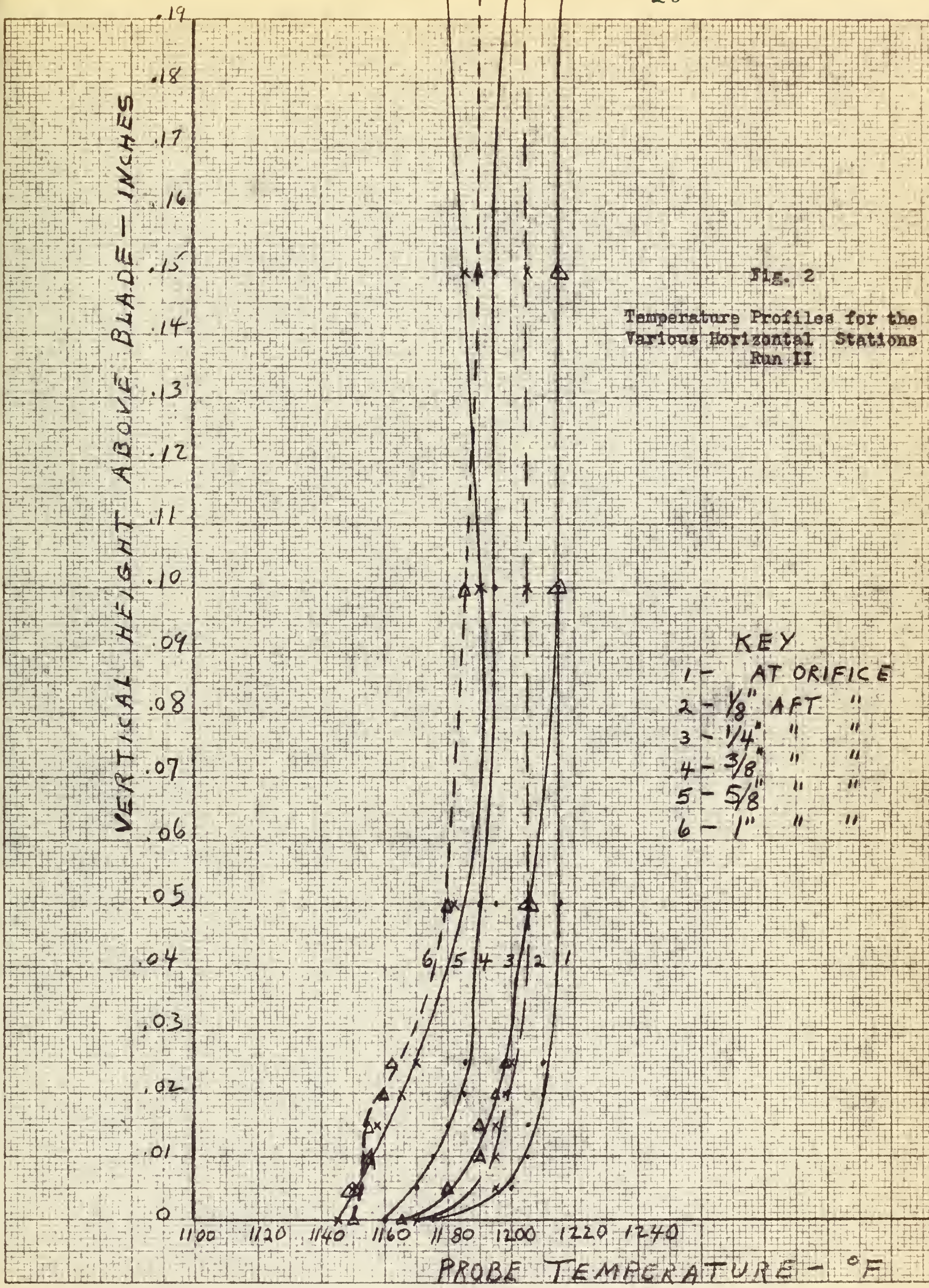


Fig. 2

Temperature Profiles for the Various Horizontal Stations Run II

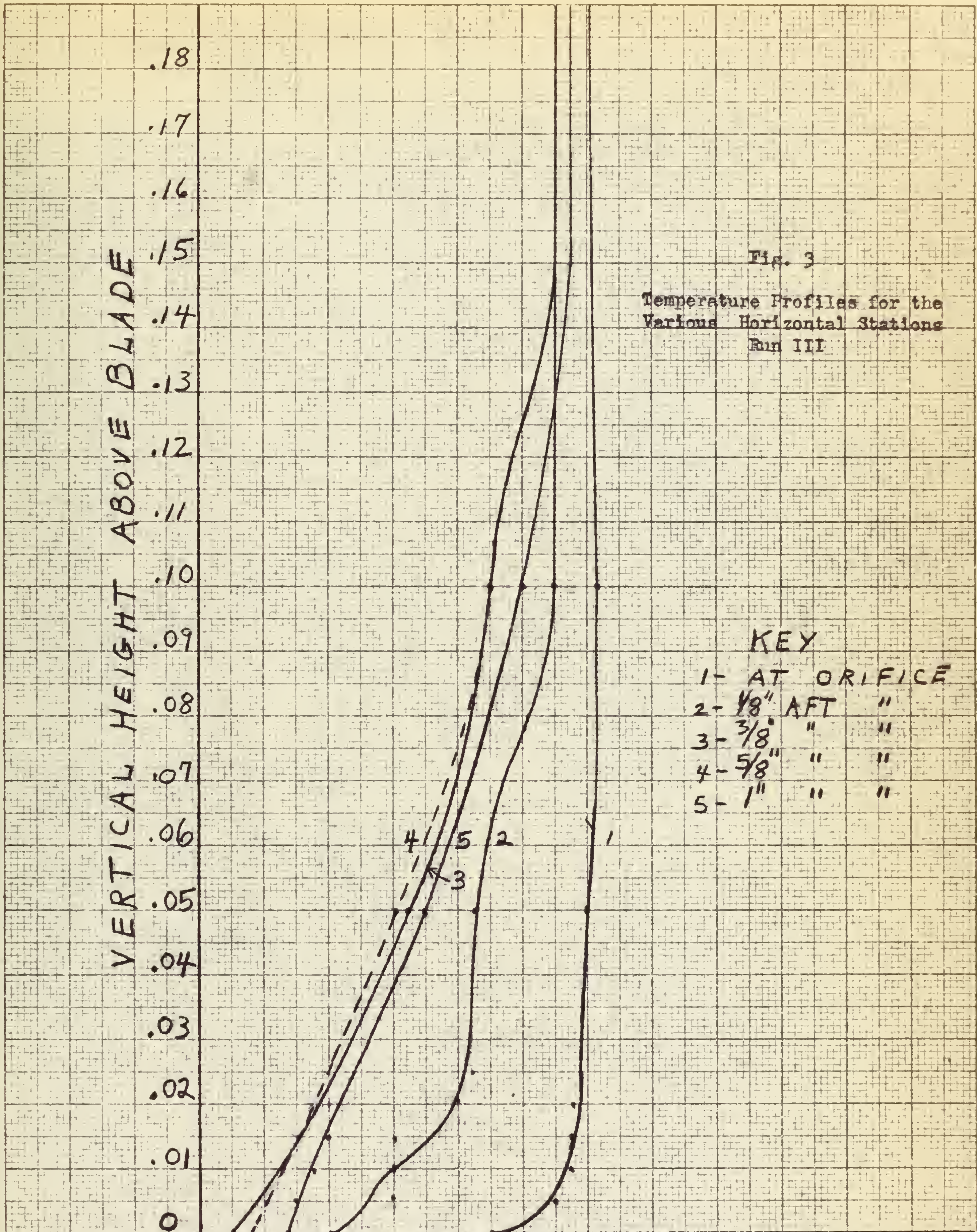


Fig. 3

Temperature Profiles for the Various Horizontal Stations Run III

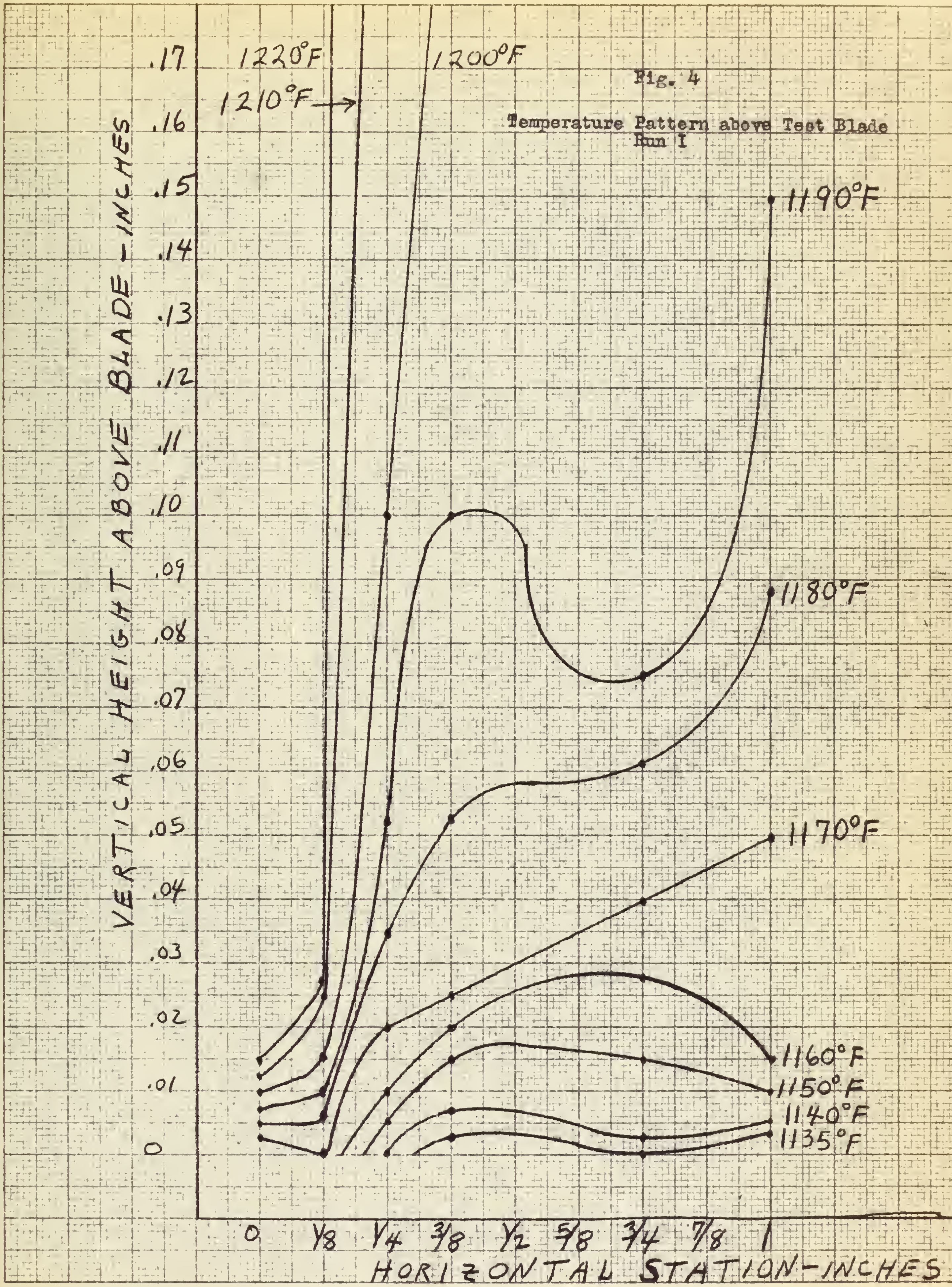


Fig. 4
Temperature Pattern above Test Blade
Run I
1220°F
1210°F
1200°F
1190°F
1180°F
1170°F
1160°F
1150°F
1140°F
1135°F
.17
.16
.15
.14
.13
.12
.11
.10
.09
.08
.07
.06
.05
.04
.03
.02
.01
0
VERTICAL HEIGHT ABOVE BLADE - INCHES
0
1/8
1/4
3/8
1/2
5/8
3/4
7/8
1
HORIZONTAL STATION - INCHES

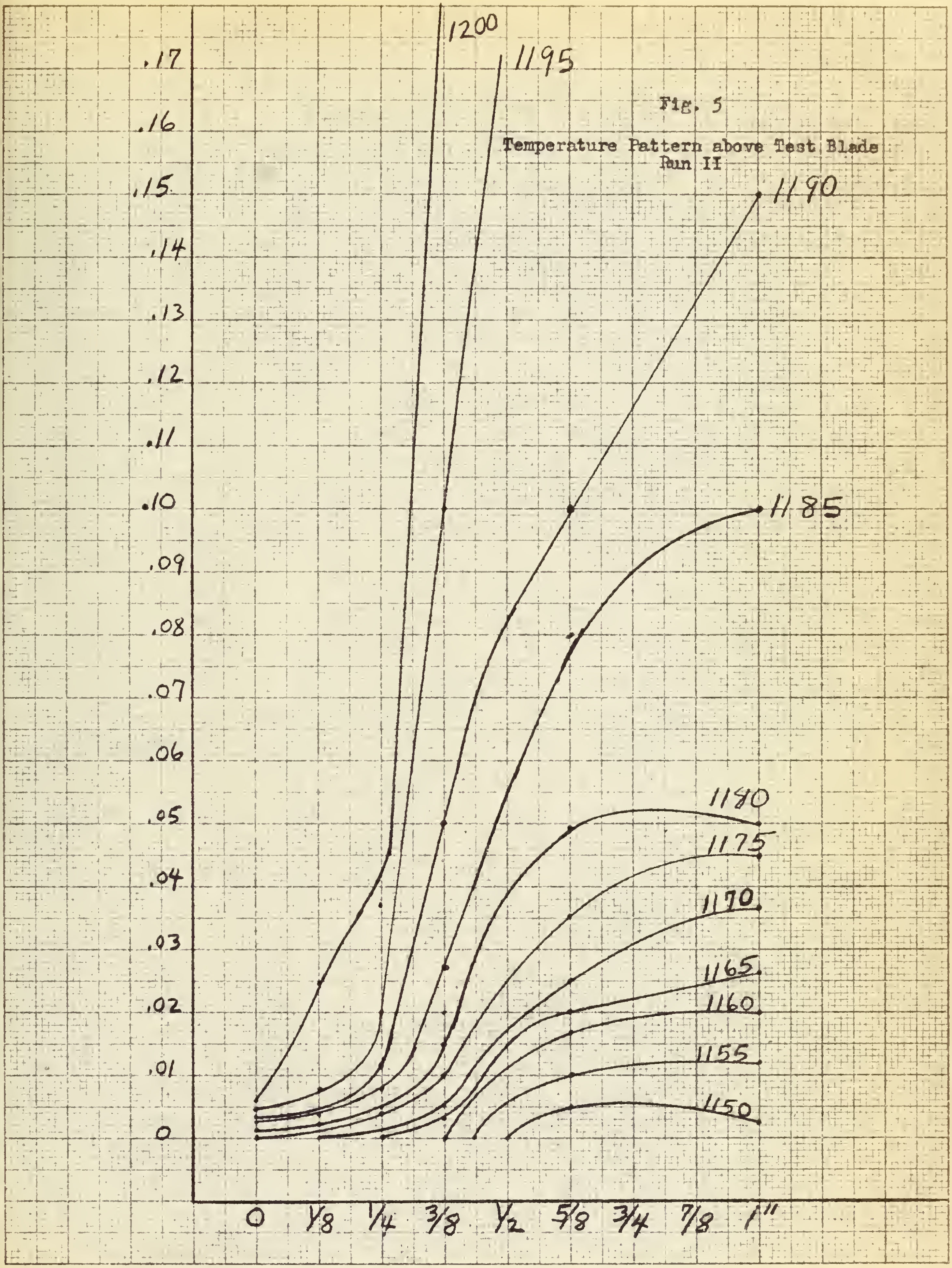
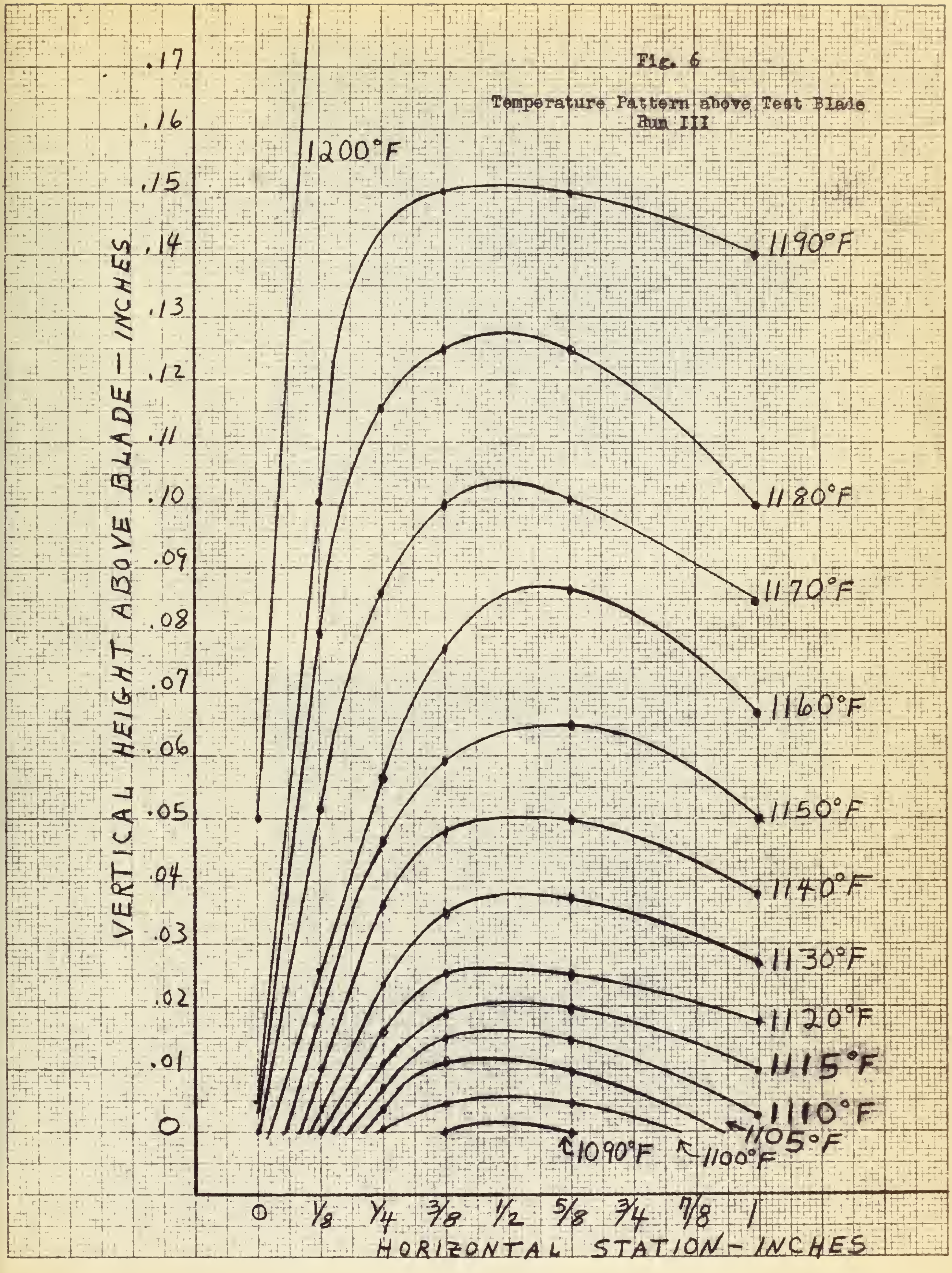


Fig. 5
Temperature Pattern above Test Blade
Run II
.17
.16
.15
.14
.13
.12
.11
.10
.09
.08
.07
.06
.05
.04
.03
.02
.01
0
0
1/8
1/4
3/8
1/2
5/8
3/4
7/8
1"
1200
1195
1190
1185
1180
1175
1170
1165
1160
1155
1150

Fig. 6

Temperature Pattern above Test Blade
Run III



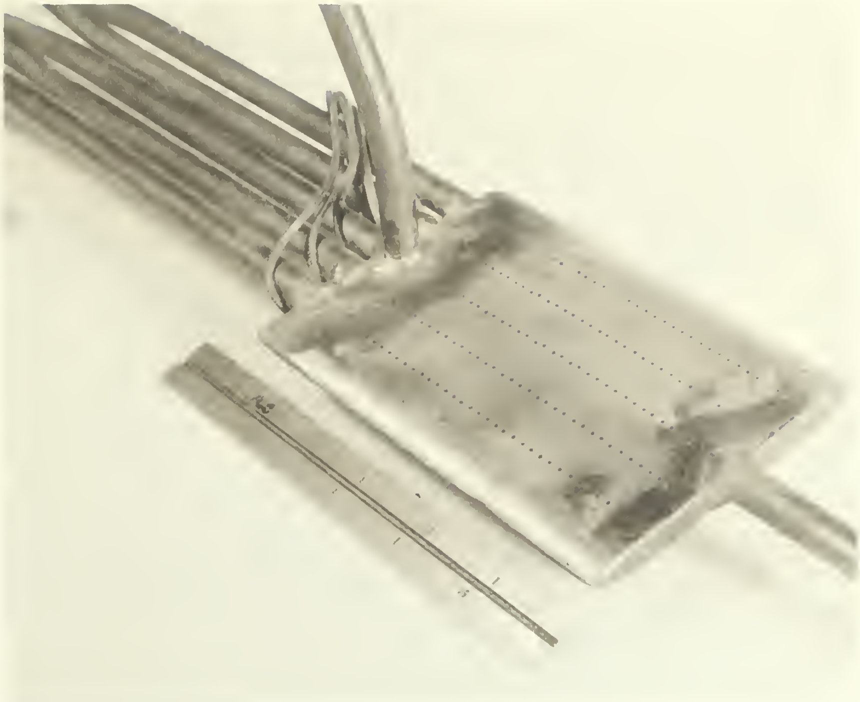


Fig. 7
TEST BLADE

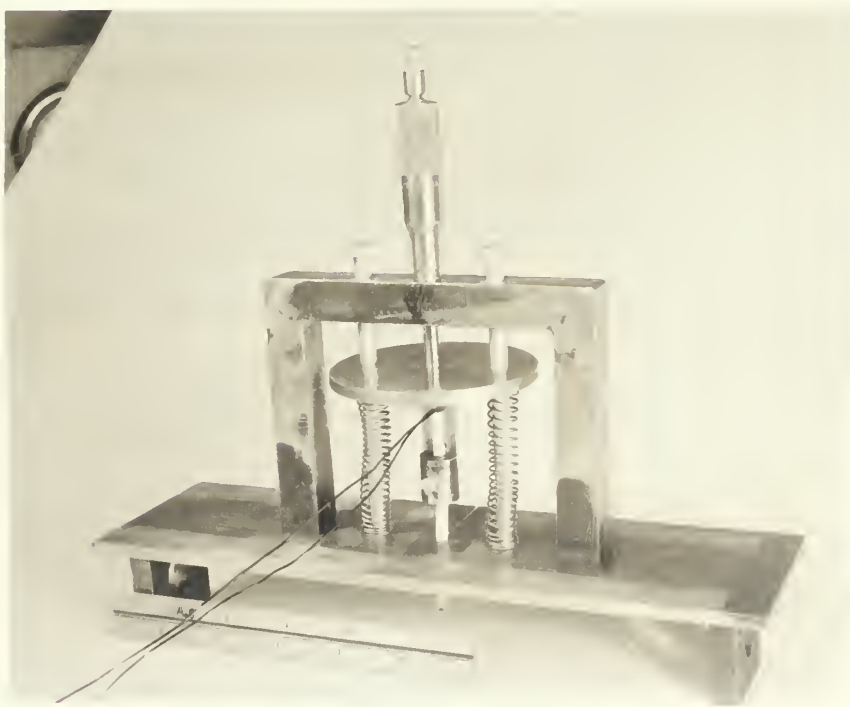


Fig. 8
TEMPERATURE SURVEY PROBE

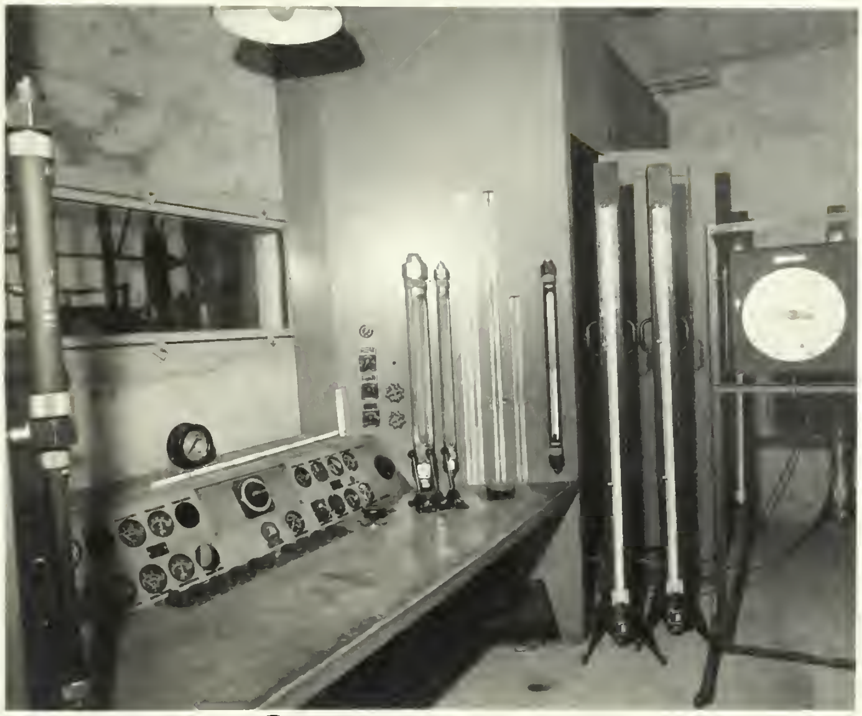


Fig. 9
Control Panel

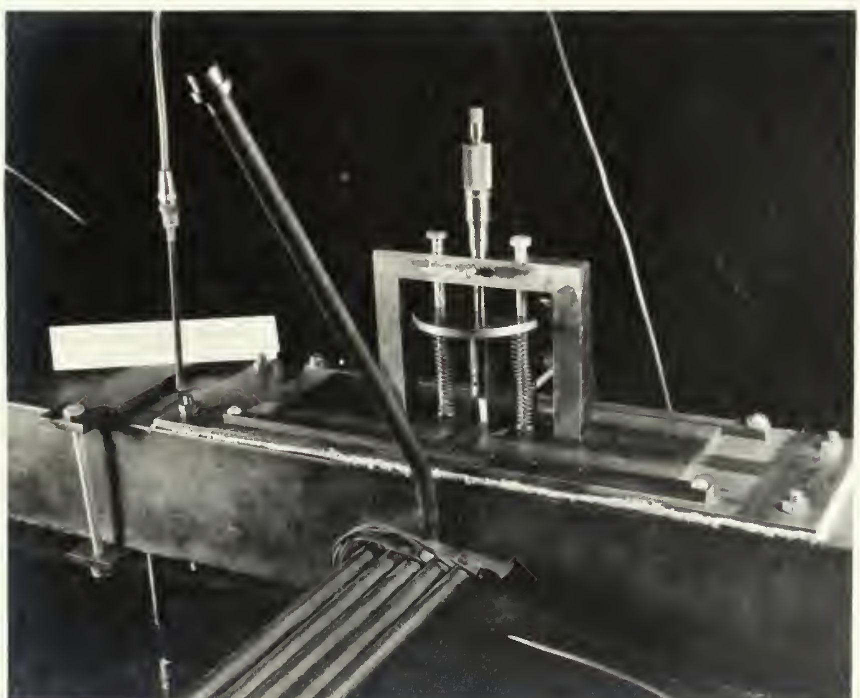


Fig. 10
Test Section

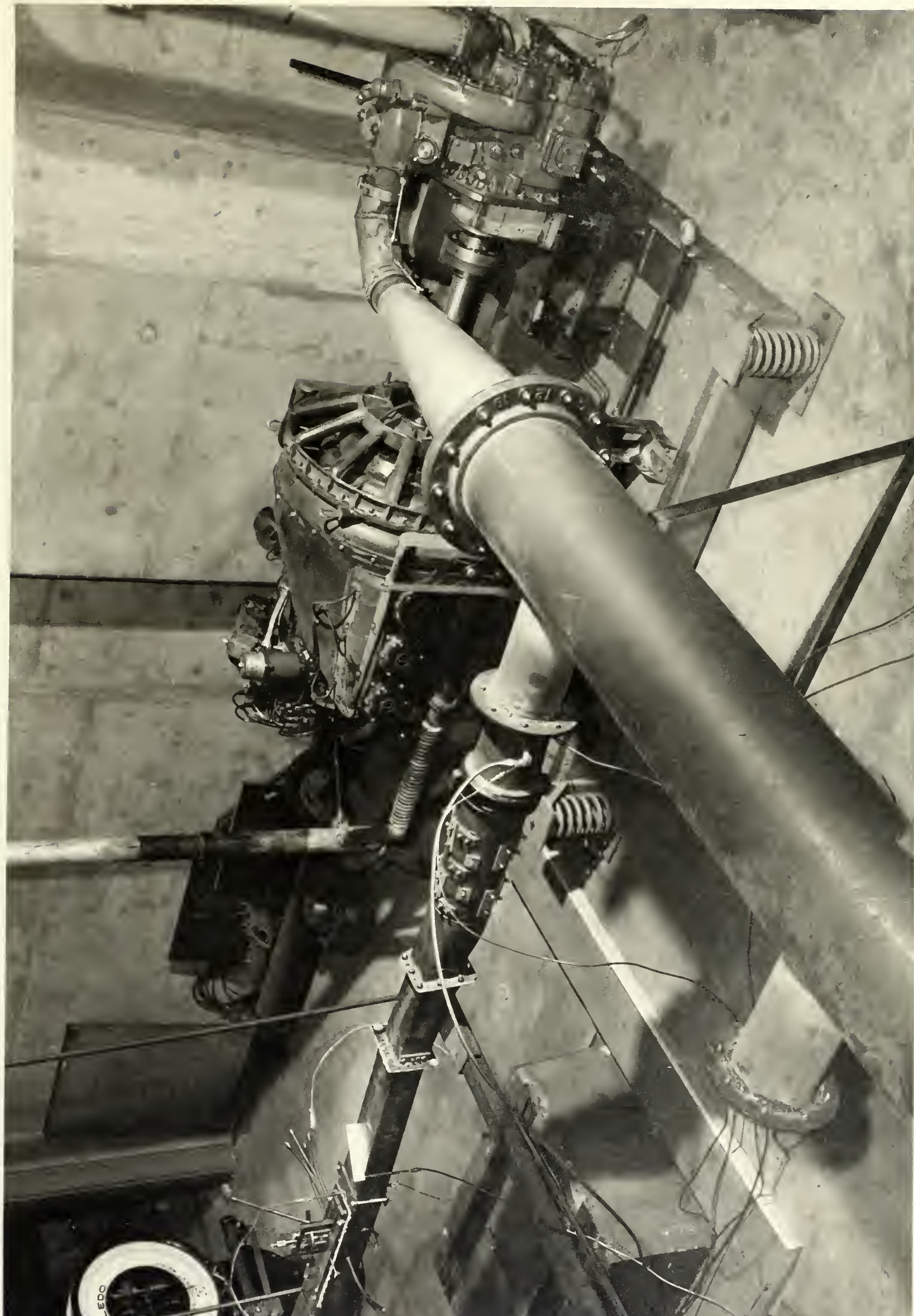


FIG. 11
TEST CELL

SAMPLE CALCULATIONS

All calculations for run #1

Burner Air Flow

$$W_{BA} = .668\, A_2 K \sqrt{\rho_1 \Delta p_{BA}} \quad \text{(Ref. 9)}$$

$$A_2 = 24.6''^2$$
$$K = .7$$

$$W = 2.52 \sqrt{\frac{(BAR)(\Delta p_{BA})}{T}}$$

$$W = 2.52 \sqrt{\frac{29.33 \times 15}{555}} = 2.24\ \text{LB/SEC}$$

Cooling Air Flow

$$Q = \frac{Q_{ROT.}}{\left[\frac{29.92}{P_{CA}} \times \frac{T}{560}\right]^{1/2}} \quad \text{(Flowrator pamphlet)}$$

$$\Delta p_{CA} = 12\ \text{psi} = 24.4''\ \text{Hg}$$
$$P_{CA} = 29.23 + 24.4 = 53.63''\ \text{Hg}$$
$$T = 95 + 460 = 555\,°R$$

$$Q = \frac{3.8}{\left[\frac{29.92}{53.63} \times \frac{555}{560}\right]^{1/2}} = 5.12\ \text{FT.}^3/\text{MIN}$$

$$5.12 \times .071\ \text{LB/FT}^3 = .363\ \text{LB/MIN.}$$

Fuel Air Ratio

$$F/A = \frac{125}{2.24 \times 3600} = .0155$$

Test section Mach

$$\frac{P_S}{P_T} = \frac{31.5}{35.53} = .887$$

For $\gamma = 1.3$ and $\frac{P_S}{P_T} = .887$, $M_{TS} = .44$ (Ref. 8)

Check on M_{TS}

$$W_{BA} = \rho_{TS} A_{TS} V_{TS}$$

$$2.24 = \frac{31.5 \times .492 \times 15.75}{1660 \times 53.3} V$$

$$A_{TS} = 15.75''^2$$

$$V = 812 \text{ FT/SEC}$$

$$T_{TS} = 1660°R$$

$$a_0 = 1950 \text{ FT/SEC}$$

$$M_{TS} = \frac{V}{a_0} = \frac{812}{1950} = .415$$

$$\% \text{ error} = \frac{.44 - .415}{.44} = 5.6\% \text{ error}$$

M_B (considering blocking effect of blade) $= .51$

Comparison to J33 turbine

$.363 \times 60 = 21.78$ LB/HR C.A. per blade

54 blades $\times$ 21.78 $=$ 1175 LB/HR Total C.A.

14 burners $\times$ $2.24 \times 3600 = 113,000$ lb/HR Total air

$$\frac{1175}{113,000} = 1.04\% \text{ Compressor Air}$$

Test Section Reynolds Number

$T = 1660°R \quad M = .44 \quad P_s = 2230 \ LB/FT.^2$

$$\rho = \frac{p}{RT} = .0253 \ LB/FT^3$$

$$\mu = 170.9 \times 10^{-6} \left[\frac{273+120}{923+120}\right]\left[\frac{923}{273}\right]^{3/2} \quad \text{(Ref. 1)}$$

$$\mu = 400 \times 10^{-6} \text{ poises} = 26.8 \times 10^{-6} \ lb/FT.SEC$$

$$Re = \frac{\rho V L}{\mu}$$

For $L = 2'' = 1/6'$

$a_0 = 1950 \ FT/SEC$

$V = M a_0 = 880 \ FT/SEC$

$$Re = \frac{(.0253)(880)(1/6)}{26.8 \times 10^{-6}} = 1.38 \times 10^5$$

Boundary Layer Thickness – (Turbulent)

$$\delta = \frac{.37 L}{\sqrt[5]{Re}} \quad \text{(Eshbach Handbook)}$$

$$\delta = \frac{(.37)(1/6)}{\sqrt[5]{1.38 \times 10^5}} = .00577' = .07''$$

Thesis 16265
T29 Terry
A temperature survey of the region aft of a cooling orifice of an air film cooled metal surface exposed to high temperature and high velocity gases.